

SPACECRAFT SYSTEM DESIGN

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The primary design objective was to maximize the probability of successful spacecraft operation within basic limitations imposed by launch vehicle capabilities, the extent of knowledge of transit and lunar environments, the current technological state of the art while providing specified payload capability. In keeping with this primary objective, design policies were established which (1) minimized spacecraft complexity by placing responsibility for mission control and decision making on earth-based equipment wherever possible; (2) provided the capability of transmitting a large number of different data channels from the spacecraft; (3) included provisions for accommodating a large number of individual commands from the earth; and (4) made all subsystems as autonomous as practicable.

The peripheral areas shown in Figure 1 represent all of the subsystems essential to the operation of the Surveyor spacecraft. These areas are discussed in subsequent sections. The Surveyor I spacecraft is an engineering test model equipped with all subsystems necessary for a soft-landing on the moon. It carries a survey television system and engineering instrument -- including temperature sensors, strain gages, accelerometers, and position-indicating devices. The orthogonal, right-hand Cartesian coordinate system has been used for this spacecraft. The configuration of the spacecraft is shown in Figure 2. A simplified functional block diagram of the spacecraft system is shown in Figure 3.

SPACECRAFT MASS PROPERTIES

Surveyor I weighed 2,192.86 pounds at launch with approximately 160 pounds separated during transit and a final touchdown weight of 610 pounds. Center of gravity limits after Surveyor/Centaur separation for midcourse and retro maneuvers are limited by the attitude correction capabilities of the flight control and vernier engine subsystems during retro-rocket burning. Limits of travel of the vertical center-of-gravity in the touchdown configuration are designed to landing site assumptions, so the spacecraft will not topple when landing.

SPACECRAFT INSTRUMENTATION

Engineering instrumentation is included on the basic bus to monitor the performance of the spacecraft and its response to the environment. The engineering instrumentation consists of the following:

- 64 resistance thermal sensors
- 12 mechanical switches
- 12 current shunts

FACILITY FORM 602

N69-75377

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(NASA CR OR TMX OR AD NUMBER)

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NONE

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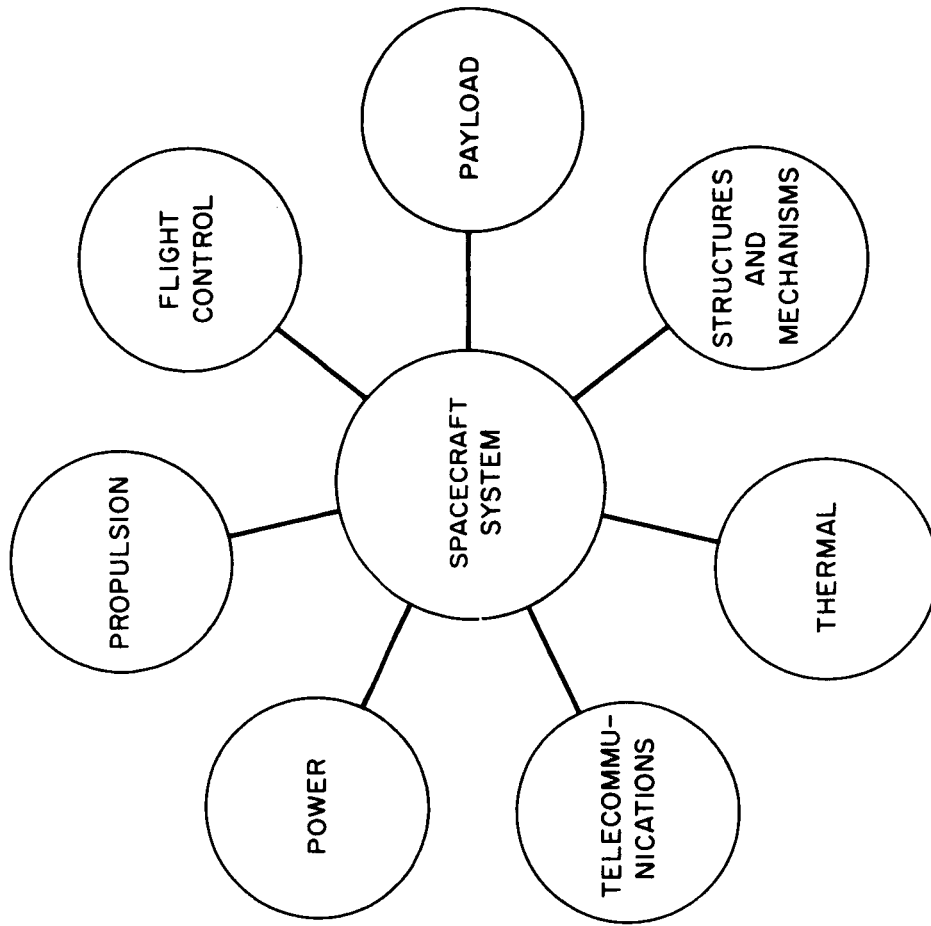


Figure 1. Spacecraft System

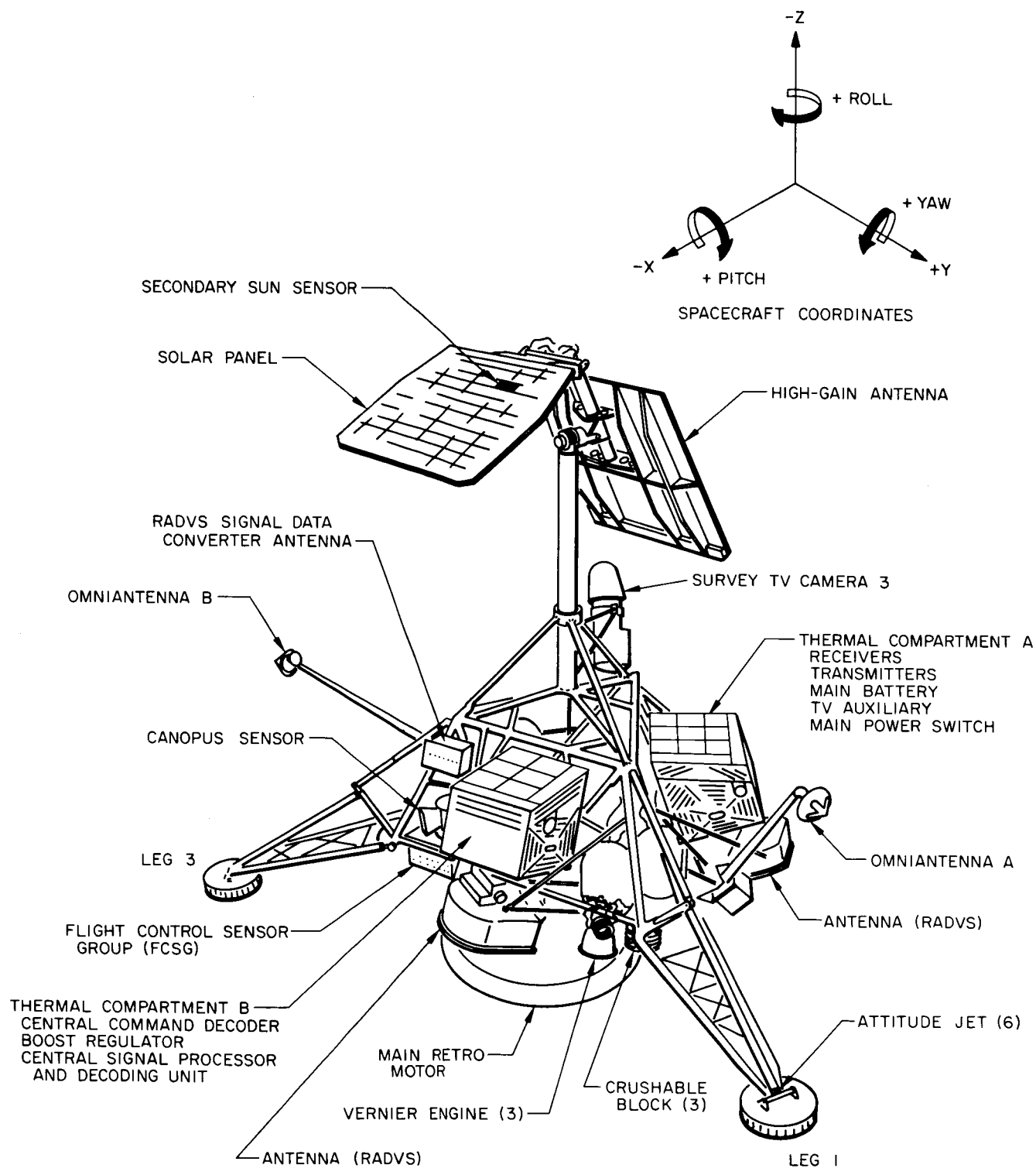


Figure 2. Spacecraft 1 Configuration

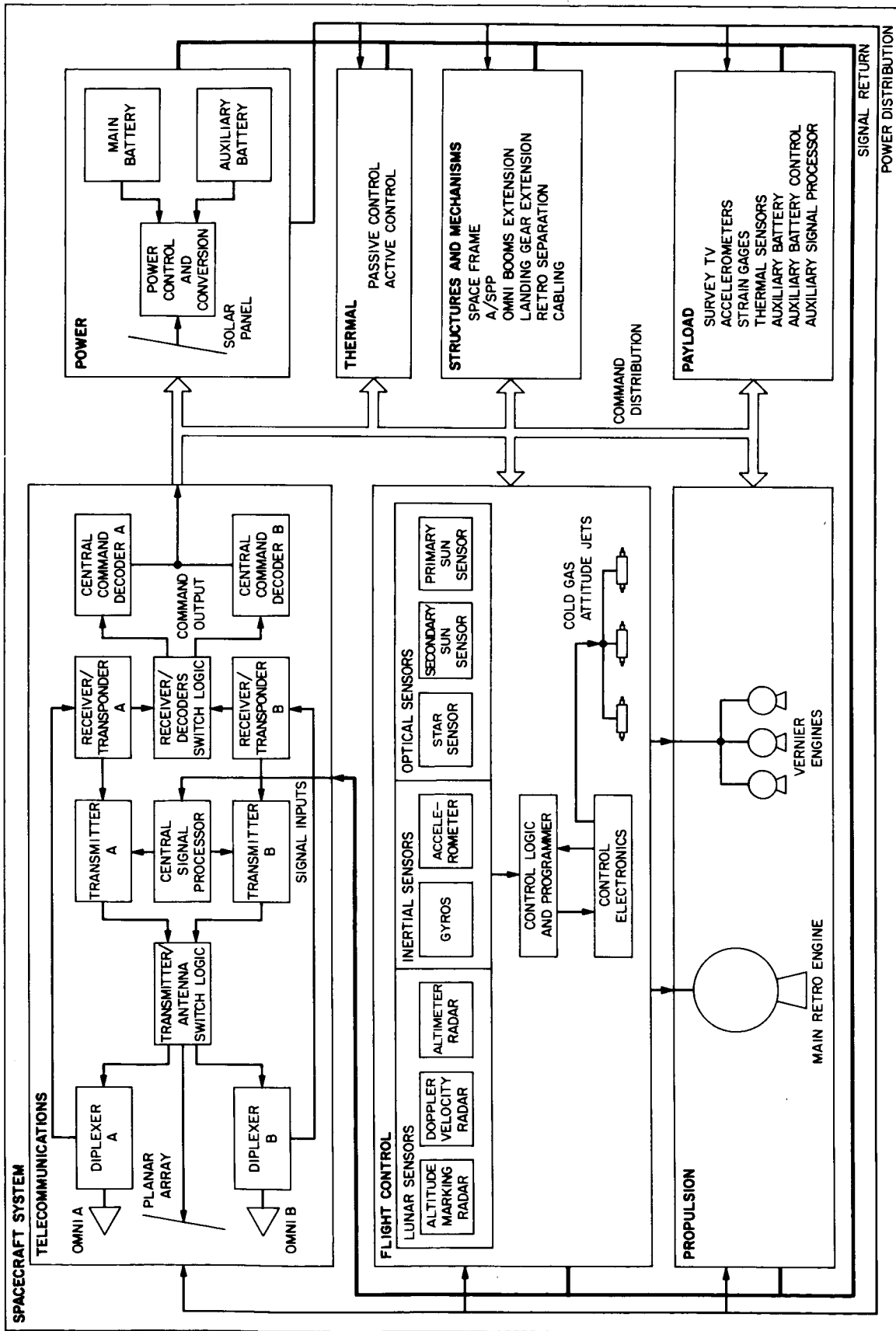


Figure 3. Simplified functional block diagram of the spacecraft system

- 7 position potentiometers
- 4 pressure sensor potentiometers
- 4 accelerometers (with one accelerometer amplifier)
- 3 spin motor running detectors
- 1 microdiode thermal sensor

THERMAL CONTROL

Thermal control of equipment over the extreme temperature range of the lunar surface (+260 to -260 F) is accomplished by a combination of passive, semipassive, and active methods. The design represents the latest state of the art in the application of structural and thermal design principles to lightweight spacecraft.

The thermal control circuits consist of heaters and heater switches that provide thermal control. Heater on-off commands are supplied by the command decoding function. Units that require thermal control are the approach and survey television cameras, AMR, compartments A and B, vernier engine No. 1 fuel and oxidizer lines, vernier engine No. 2 fuel and oxidizer tanks and lines, and vernier engine No. 3 oxidizer tank and fuel and oxidizer lines.

STRUCTURES AND MECHANISMS

The structures and mechanisms subsystem provides support, alignment, thermal protection, electrical interconnection, mechanical actuation, and touchdown stabilization for the spacecraft and its components. The subsystem includes the basic spaceframe, landing gear mechanism, crushable blocks, omnidirectional antenna mechanisms, antenna/solar panel positioner, pyrotechnic devices, electronic packaging and cabling, thermal compartments, thermal switches, separation sensing and arming device, and secondary sun sensor.

The mechanical and thermal services function consists of pyrotechnic devices, mechanical devices, and electronic and thermal control circuits. Its purpose is to actuate and/or release mechanical devices, turn the RADVS on and off, position the planar array antenna and solar panel, and provide thermal control for spacecraft control items.

The mechanical and thermal services function requires +22 v and +29 v power from the power management function and +4.85 v from the signal processing function. Commands from ground control provide power switching and activation of the individual circuits.

FLIGHT CONTROL

The flight control function consists of equipment within the flight control sensor group, secondary sun sensor, attitude jets, roll actuator, planar array, altitude marking radar, RADVS items, vernier engine items, and the retro rocket.

The purpose of flight control is to control spacecraft flight parameters throughout the transit portion of the mission. Flight control uses three forms of reference to perform its mission. These are celestial sensors, inertial sensors, and radar

sensors. The outputs of each of these sensors are utilized by the analog electronics to create thrust commands which operate the propulsion systems on board the spacecraft. Flight control programming initiates and controls sequences within the remainder of the function. The propulsion systems are the vernier engines; the attitude gas jets; and the retro rocket, which is ignited in response to a signal from a switch in the mechanical and thermal services function.

Power for most of the flight control electronics is supplied by 22 v unregulated power from the mechanical and thermal services function and 29 v flight control power from the power management function. The RADVS section requires the 22 v output of a pyro switch in mechanical and thermal services. In addition, flight control requires ground commands that initiate various sequences and perform manual operations.

The celestial sensors allow the spacecraft to be locked to a specific orientation defined by the lines to the sun and the star Canopus and the angle between them. Initial search and acquisition of the sun is accomplished by the secondary sun sensor. The primary sun sensor then maintains the orientation with the sun line.

The integrating gyros provide for maintaining spacecraft orientation inertially when the celestial references are not available. The accelerometers measure the thrust levels of the spacecraft propulsion during midcourse correction and retro descent phases.

The altitude marking radar provides a trigger pulse to initiate the retro staging sequence.

The RADVS functions in the flight control subsystem to provide three-axis velocity, range, and altitude mark signals for flight control during the retro and vernier phases. The RADVS consists of a doppler velocity sensor, which computes velocity along each the S/C X, Y, and Z axes, and a radar altimeter, which computes slant range from 40,000 ft to 13 ft and generates 1000 ft and 13 ft mark signals. The RADVS comprises four assemblies: 1) klystron power supply/modulator (KPSM), which contains the RA and DVS klystrons, klystron power supplies, and altimeter modulator; 2) altimeter/velocity sensor antenna, which contains beams 1 and 4 transmitting and receiving antennas and preamplifiers; 3) RADVS velocity sensing antenna, which contains beams 2 and 3 transmitting antennas and preamplifiers; 4) RADVS signal data converter, which consists of the electronics to convert doppler shift signals into dc analog signals.

The RADVS accepts +22 v unregulated from the "pyro switch output 22 v" line from the engineering mechanisms auxiliary unit. A pyrotechnic switch in the EMA turns the RADVS power on at about 50 miles and off at about 13 ft.

The attitude jets are cold gas reaction devices which control the orientation of spacecraft attitude in all coordinates during coast phases of the flight. They are installed in opposing pairs on the ends of the landing gear.

The vernier engines supply a continuously variable thrust for control of the spacecraft velocity vector and controlled descent to the lunar surface. The roll actuator tilts the thrust axis of thrust chamber No. 1 away from the spacecraft roll axis for attitude and roll control during thrust phases of flight, i. e. when the vernier engines are operating and the attitude jets are not.

The retro rocket removes the major portion of the spacecraft approach velocity during descent. It is triggered by squibs which fire upon signal from the programmer via squib firing circuits in mechanical and thermal services.

ELECTRICAL POWER

The electrical power subsystem is designed to generate, store, convert, and distribute electrical energy. The subsystem can generate a continuous unregulated power of 72 to 56 watts depending on the environmental conditions. Its peak unregulated power capability is 1000 watts, limited in time by the energy stored. The initial energy storage of the subsystem is 4730 watt-hours provided by two batteries. Only one battery, the main battery, can be recharged to an energy storage of 3520 watt-hours. The batteries determine the unregulated power voltage and are designed to sustain a voltage between 17.5 and 27.5 volts with a nominal value of 22 volts. The unregulated power is distributed to the unregulated loads and regulated loads via the unregulated bus.

Regulated power is provided by the boost regulator at 29.0 volts, controlled to one percent for the flight control and nonessential loads and to two percent for the essential loads. The maximum regulated power capability of the boost regulator is 270 watts.

TELECOMMUNICATIONS

Communication with the spacecraft is accomplished by the RF (radio frequency) link. Critical operations such as midcourse maneuver and descent phases are timed to coincide with control by the Goldstone DSS. Two identical receivers (A and B) aboard the spacecraft receive commands from ground control. Both receivers operate continuously throughout the life of the spacecraft to provide continuous command capability. The spacecraft processes data produced by the voltage, current, temperature, and pressure sensors as well as the television camera, accelerometers, and strain gages. Such data are transmitted to ground stations by two identical transmitters (A and B) on board. Transmitter power at the antenna varies from a nominal 100 milliwatts (low power) to 10 watts (high power).

The number of telemetry data channels operating simultaneously during the mission is controlled by ground commands in accordance with the phase of each mission. There are seven commutated channels, as shown in Table 1. The commutators can be operated upon earth command at 17-3/16, 137-1/2, 550 1100, and 4400 bits/sec. Choice of the bit rate is determined by the strength of signals received at the DSIF station.

The spacecraft is equipped with three antennas, two omni antennas and the planar array. All of the antennas are capable of operating with the transmitters in the high or low power mode. The planar array is required for efficient transmission of television signals. Switching allows the use of alternate antennas with the transmitters.

Receiver/transponder A and receiver/transponder B are identical. Each is directly connected to one of the omni antennas. The receiver/transponders also provide a phase-coherent system for doppler tracking from the DSIF.

TABLE 1. CONTENT OF TELEMETRY SIGNALS FROM SPACECRAFT

Data	Source	Significance	Number of points sampled	Form	Comments
Commulator, Mode 1	Flight control, propulsion	Provides data required for midcourse maneuver	100	Digital	Modes 1, 2, 3, and 4 used one at a time on command per Standard Sequence of Events (SSE)
Commulator, Mode 2	Flight control, propulsion, approach TV, AMR, RADVS	Provides data required for retro descent	100	Digital	
Commulator, Mode 3	Inertial guidance, approach TV, AMR, RADVS, vernier engines	Provides data required for vernier descent	50	Digital	
Commulator, Mode 4	Temperatures, power status, telecommunications	Provides data required for miscellaneous transit and lunar surface operations	100	Digital	
Cruise phase commulator, Mode 5	Flight control, power status, temperature	Provides data required during cruise mode to determine general spacecraft status	120	Digital	Used on command per SSE
Thrust phase commulator, Mode 6	Flight control, power status, AMR, RADVS, vernier engine conditions	Provides data required for backup of Modes 1, 2, and 3 during thrusting maneuvers	120	Digital	Used on command per SSE
TV commulator, Mode 7	TV survey camera	Provides frame identification while survey TV is operating	16	Digital	Frame ID alternates with analog video signals
Vibration	Accelerometers	Indicates vibrations due to booster engine, main retro-rocket, and vernier engine firings and mechanical shock during landing	8	Analog	
Shock absorber data	Strain gages	Measures strain on landing gear due to landing shock	3	Analog	
Gyro speed	Inertial guidance unit	Indicates angular rate of gyro spin motors	3	Analog	Samples are pitch, roll, and yaw axes on command per SSE

The receivers consist of two identical receiver-transponder units, A and B, each composed of solid state devices, cavities and attenuators. The purpose of the receivers is to receive commands from ground control and to provide for doppler tracking (transponder operation) of the spacecraft while it is in transit.

Both receivers operate continuously throughout the life of the spacecraft to provide constant command capability. The outputs of both receivers are connected to the central decoder unit (CDU). Each receiver is permanently connected to an omni directional antenna through a diplexer. Transponder interconnection circuits are provided that permit each receiver to be connected to a transmitter for transponder operation wherein the transmitted signal is in phase coherence with the received signal.

The receivers normally operate as crystal-controlled double-conversion frequency-modulation receivers. Commands from ground control are frequency modulated on a subcarrier which in turn phase modulates the main carrier of the earth-to-spacecraft transmissions. The detected signal is then applied to a sub-carrier discriminator which recovers the digitized command signals and supplies them to the command decoding function.

Receiver/transponder A and B are activated prior to launch by application of "+22 V Unreg" permitting the reception of commands from ground control. The commands are sent by ground control to activate and check the operation of the various systems within the spacecraft prior to launch and throughout the life of the spacecraft.

The transponders consist of two identical transponder interconnections, each composed of solid state circuits located in receiver/transponder A and receiver/transponder B. The purpose of the transponder, operating in conjunction with a receiver and a transmitter, is to provide a phase coherent system for doppler tracking of the spacecraft during transit. There are two complete receiver-transponder-transmitter systems to ensure reliability.

The combination of transmitter B and receiver/transponder B is normally operated as a transponder during transit by means of the transponder connections in receiver/transponder B. In the event of a failure in the B system, another identical set of transponder interconnections may be used to connect transmitter A and receiver/transponder A for transponder operation.

The command decoding function receives the commands detected by the RF data link function. The commands, "receiver A output" and "receiver B output", are used by the two central command decoders to generate sync and timing signals.

The commands are also supplied to the receiver/decoder selector which selects the combinations of receivers and central command decoders.

If either receiver fails to supply output signals, the receiver/decoder selector changes the receiver-decoder combination. Altogether, four combinations are possible. Selection can be made by interrupting the flow of messages to the spacecraft.

The subsystem decoders receive the direct command outputs of the central command decoders; they supply specific commands to the other functions of the system.

The signal processing function gathers data produced by the other areas of the system and prepares the data for transmission by the RF data link to the DSIF. The information is that necessary to monitor all of the spacecraft functions; the majority of the information has been preprocessed to be within the range of 0 to 5 vdc.

The commutation section divides the data signals into several groups. The signals in each group, called a commutator mode, are sent on a time-shared basis, each signal having one or more places in a sequence. The particular mode to be telemetered is commanded by the DSIF.

The transmitter planar antenna control is a control that allows signals from transmitter A or B to be sent to the planar array antenna. When the "xmtr B to planar" signal is present, a +22 v "ant. xfer sw, xmtr B to planar" signal is sent to the antenna transfer switch. When the "xmtr A to planar" signal is present, a +22 v "ant xfer sw, xmtr A to planar" signal is sent to the antenna transfer switch. The "ant. xfer sw, xmtr B to planar" signal connects the "xmtr B RF output" from transmitter B to the planar array antenna and transmitter A to the omni antenna A or B. The "ant xfer sw, xmtr A to planar" signal connects the "xmtr A RF output" from transmitter A to the planar array antenna and transmitter B to omni antenna A or B.

The diplexer permits simultaneous transmission and reception of different frequencies on the same antenna. The transmitter RF output from the diplexer is applied to the stripline power monitor. A small portion of the transmitter RF, during high power transmission only, is rectified and filtered by the stripline power monitor and supplied to ESP voltage measurement commutation that is transmitted to ground control as an indication of transmitter operation. The remaining portion of the transmitter RF output is applied through the stripline power monitor to omni antenna B and radiated to ground control.

The transmitter function consists of two identical transmitters, A and B. The transmitters may be switched so that either may use the high-gain planar array or the omni-directional antennas, providing a complete and redundant transmitter function. Each transmitter consists of solid state circuits, mechanical switches, attenuators, isolators, and a traveling-wave tube (TWT). The purpose of this function is to transmit the following information to the FSIF:

1. Engineering data during transit and lunar phases.
2. Scientific data during lunar operations.
3. Verification of commands received from the DSIF during transit and lunar phases.
4. Doppler tracking signal during transit.
5. Television data during descent and lunar phases.

Transmitter A is used as the standby transmitter and is activated when transmitter B fails. Transmitter low power must be turned on before high power can be turned on. Transmitter B is activated prior to launch by application of commands from subsystem decoder No. 1, which permits the transmission of engineering data and verification of commands received to ground control while the spacecraft is still on the launch pad. This information is required by ground control during operational

checkout of the Surveyor system. The transmitters may be used as long as the "+29 V regulated non-essential" power is available. The "+22 V unregulated" is required to operate the relays for switching between high and low power operation and for the antenna transfer switches.

The transmitters are capable of operating in several modes during the functional modes of transit. The different operating and functional modes are as follows:

1. High power mode.
2. Normal mode.
3. Emergency mode.
4. Transponder mode.

PROPULSION

The propulsion subsystem supplies thrust force during the midcourse correction and terminal descent phases of the mission. The propulsion subsystem consists of a vernier engine system and a solid propellant main retro rocket engine. The propulsion subsystem is controlled by the flight control system through preprogrammed maneuvers, commands from earth, and maneuvers initiated by flight control sensor signals.

The vernier propulsion subsystem supplies the thrust forces for midcourse maneuver velocity vector correction, attitude control during main retro rocket engine burning, and velocity vector and attitude control during terminal descent. The vernier engine system consists of three thrust chamber assemblies and a propellant feed system. The feed system is composed of three fuel tanks, three oxidizer tanks, a high-pressure helium tank, propellant lines, and valves for system arming, operation, and deactivation. The thrust of each engine (which is monitored by strain gages installed on each engine mounting bracket) can be throttled over a range of 30 to 104 pounds. The specific impulse varies with engine thrust.

The main retro-rocket, which performs the major portion of the deceleration of the spacecraft during lunar landing maneuver, is a spherical, solid-propellant unit with a partially-submerged nozzle to minimize over-all length. The engine utilizes a PBAA composit-type propellant and conventional grain geometry.

The motor case is attached at three points on the main spaceframe near the landing leg hinges, with explosive nut disconnects for post-burnout ejection. Friction clips around the nozzle flange provide attachment points for the altitude making radar. The retro-rocket, including the thermal insulating blankets, weighs approximately 1390 lbs. This total includes about 1244 lbs. of propellant. The thermal control design of the retro-rocket engine is completely passive, depending on its own thermal capacity, insulating blankets (21 layers of aluminized mylar plus a cover of aluminized Teflon), and a surface coating. The prelaunch temperature of the unit is $70 \pm 5^\circ\text{F}$. At terminal maneuver, when the engine is ignited, the propellant will have cooled to a thermal gradient with a bulk average temperature of about 50 to 55°F .

The altitude marking radar (AMR) normally triggers the terminal maneuver sequence. When the retro firing sequence is initiated the retro-rocket ignitor gas

pressure ejects the AMR. The engine operates at a thrust level of 8,000 to 10,000 lb. for approximately 39 seconds at an average propellant temperature of 50°F.

PAYLOAD

The engineering payload provides telemetry data for evaluation of the spacecraft status and its capabilities for performing a lunar mission. Data to be provided includes vernier engine thrust, main retro engine case pressure, touchdown shock, and thermal status of critical components of the system including the structure. The engineering payload consists of the following:

- Survey television camera.

- Television auxiliary.

- Auxiliary engineering signal processor.

- Auxiliary battery.

- Auxiliary battery control.

- Engineering payload instrumentation sensors.

The survey television provides pictures of the lunar surface, portions of the spacecraft, and free space. The survey television is used during lunar operations. The auxiliary engineering signal processor processes engineering payload data for telemetry to DSIF. The auxiliary battery provides a back-up power source for the main battery and solar panel and the auxiliary battery control provides for automatic and command controlled application of the auxiliary battery. The engineering payload instrumentation provides additional data on the performance status of the spacecraft and its response to the environment, beyond the capacity of the basic bus engineering instrumentation. The engineering payload instrumentation sensors consist of the following:

- 11 resistance thermal sensors.

- 7 strain gages (with one strain gage amplifier assembly).

- 4 accelerometers.

A strain gage is mounted on each of the three shock absorbers to monitor shock absorber loads during lunar touchdown. A strain gage is also mounted on each of the three vernier engine brackets to provide correlation as to vernier engine response to flight control system commands and yield gross indication of engine thrust via the strain induced by engine operation. A strain gage on the main retro rocket engine case verifies that the engine has ignited and provides a gross indication of engine performance.

The four accelerometers are located in compartment A and B, one on the A/SPP mast, and one on the RADVS velocity sensing antenna. Full-scale range of the accelerometer-amplifier system is ± 15 g (high-gain output). The accelerometers indicate vehicle and component accelerations during all mission stages after DSIF acquisition.

Survey television camera provides the capability of observing the lunar surface, portions of the spacecraft, and large sections of free space, on command from earth. The camera can be commanded to alter its angular field of view and to change the angular orientation of the center of the field of view with respect to the basic spacecraft coordinate system. Provisions are made for inserting colored or polarizing filters into the camera optical system on command from earth. The focus distance and lens aperture are adjustable on command from earth for variation in object distance and light intensity. Provision is also made to alter the lens opening (iris) either on direct command from earth or automatically as desired. Temperature sensing devices are also contained on the camera which give the temperature conditions.

Survey Television Camera converts optical images within its field of view into complete composite video signals which include horizontal and vertical synchronization and vertical blanking pulses. In performing this function, the camera is completely self-contained, requiring only electrical power inputs, and decoded commands from earth.

Survey television camera may be operated in a 600 TV line normal mode or a 200 TV line emergency mode, the normal mode providing better quality at the expense of greater signal bandwidth. Two major command signals are transmitted to the camera during its operational sequence. "Survey camera, power on," and "start frame" commands. Additional commands are transmitted to the camera allowing remote control of various mechanisms.

SPACECRAFT RELIABILITY

The prelaunch reliability estimate for Surveyor I was 0.51 for the flight and landing mission. The estimate is based on systems test data. Owing to the number of unit changes on the spacecraft, the reliability estimate is considered generic to Surveyor I rather than descriptive of the exact Surveyor I spacecraft configuration. Figure 4 shows the history of reliability estimates for Surveyor I during its system test phases.

The detail reliability estimates for flight and landing are listed in Table IV-2.

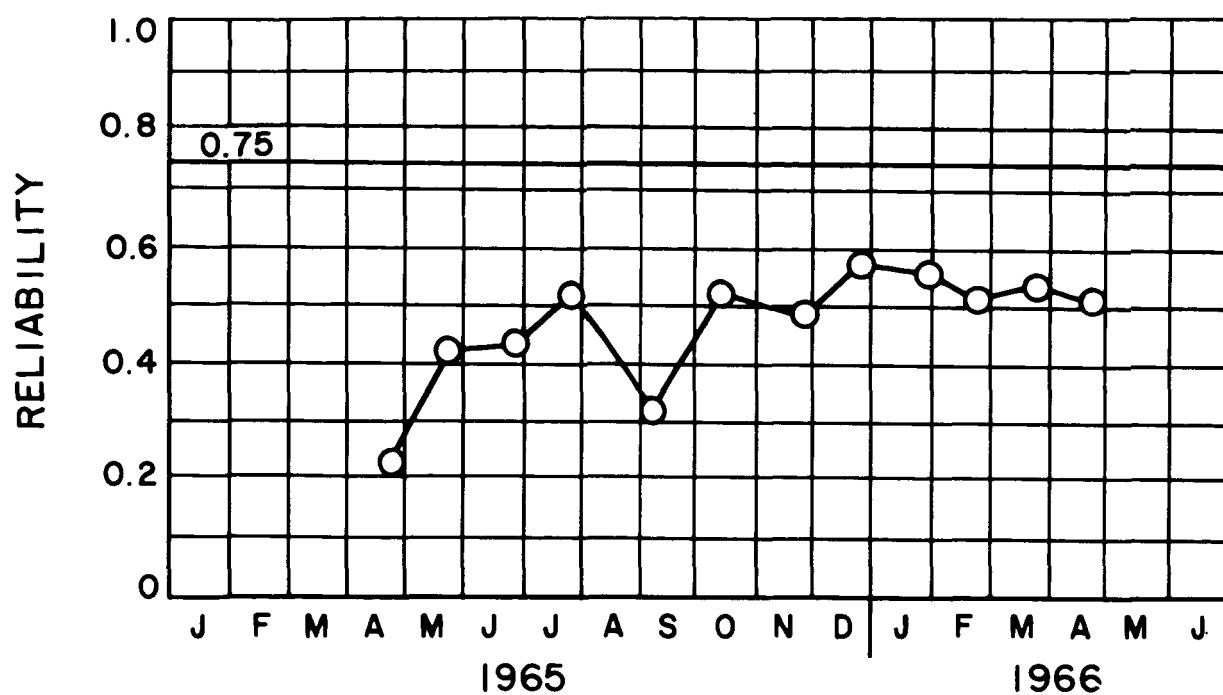


Figure 4. Spacecraft 1 system test reliability history

TABLE 2. SURVEYOR I SPACECRAFT RELIABILITY
(flight and landing)

Subsystem	Reliability estimates
Telecommunications	0.922
Vehicle mechanisms	0.854
Propulsion	0.991
Electrical power	0.866
Flight controls	0.954
Spacecraft	0.645
(System interaction reliability factor)	(0.788)
Spacecraft reliability = (0.645) (0.788) = 0.51	